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EFFECT OF MACH AND REYNOLDS NUMBERS ON THE MAXIMUM LIFT COEFFICIENT OBTAINABLE IN GRADUAL AND ABRUPT STALLS OF A PURSUIT AIRPLANE EQUIPPED WITH A LOW-DRAG WING

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MEMORANDUM REPORT

for the

Bureau of Aeronautics, Navy Department

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SUMMARY

Flight tests were conducted on a pursuit airplane, which has an NACA low-drag wing, to determine the effects of Mach and Reynolds numbers on the maximum lift coefficient obtainable in gradual and abrupt stalls. Gradual stalls were made at Mach numbers from 0.145 to 0.67 and Reynolds numbers from 5,200,000 to 19,300,000. Stalls of varying degrees of abruptness were made at selected Mach numbers from 0.195 to 0.44 and Reynolds numbers from 6,370,000 to 11,300,000.

The test results indicated that the maximum lift coefficient obtainable in a gradual stall was greatly affected by Mach number as well as Reynolds number, even when the Mach number was as low as 0.15. As the Mach number was increased from the lowest value tested, the maximum lift coefficient decreased steadily until a minimum value of 0.90 was reached at a Mach number of 0.49 and then increased reaching a value of 1.09 at a Mach number of 0.66. In addition, the usual Reynolds number effects were postponed to a larger Reynolds number and were diminished in magnitude until, at speeds greater than the critical Mach number, no effects of Reynolds number were apparent. The maximum lift coefficients obtainable in abrupt stalls were found to be limited by Mach number but were independent of Reynolds number.

A comparison is made with corresponding data obtained for a pursuit airplane equipped with a conventional wing, which is very similar to the test airplane. Results show that at low Mach numbers the maximum lift of the airplane with the conventional wing was greater than that of the airplane equipped with the low-drag wing. At moderately supercritical Mach numbers, however, the maximum lift of the airplane with the low-drag wing was much greater than that of the airplane with the conventional wing.

INTRODUCTION

While extensive information on the characteristics of NACA low-drag airfoils has been provided by wind-tunnel tests, the characteristics of practical construction wings having the low-drag airfoil sections have not been as completely evaluated. In order to provide comparative data on the maximum lift characteristics of wings having low-drag and conventional airfoil sections, flight tests have been conducted at the Ames Aeronautical Laboratory on two airplanes of otherwise similar configuration.

Reference 1 presents data on the effects of Mach and Reynolds numbers on the maximum lift coefficient obtainable in gradual stalls of a pursuit airplane which has a wing with conventional airfoil sections. In the present report information is provided on the effects of Mach and Reynolds numbers on the maximum lift coefficient obtainable in gradual and abrupt stalls of a similar pursuit airplane which has a wing with low-drag airfoil sections. For purposes of comparison data from reference 1 are included in the present report.

DESCRIPTION OF THE AIRPLANE

The test airplane is a single-place, single-engine, low-wing, cantilever monoplane. Figure 1 is a three-view drawing of the airplane and figure 2 shows the airplane as instrumented during the tests. The profiles of the root and tip airfoils are shown in figure 3. The general specifications of the airplane are as follows:

Engine . . . . Liquid-cooled, V-12, V-1710-93

Propeller. . . . . Hydraulically operated, constant speed

Diameter. . . . . 11 ft 7 in.

Propeller design. A642-S-D1

Blade design. A-20-156-17

Weight at take-off (as flown). . . . . 8300 lb

Center-of-gravity position at take-off (as flown). . . . . 26.3 percent M.A.C.

Wing

Span. . . . . 38.33 ft

Area. . . . . 248 sq ft

Aspect ratio. 5.93

Taper ratio 2:1

Incidence, root . . . . $1^\circ 18'$

Incidence, tip. . . . . $-0^\circ 27'$

Dihedral (top surface, 35 percent chord). . . . . $3^\circ 40'$

Sweepback (leading edge). . . . . $5^\circ 6' 17''$

M.A.C. . . . . 6.88 ft

Airfoil, root . . . . NACA 66,2X-116(a=0.6)

Airfoil, tip. . . . . NACA 66,2X-216(a=0.6)

Horizontal-tail surfaces

Incidence . . . . 1°

Total area . . . . 45.15 sq ft

INSTRUMENTATION

Standard NACA photographically recording flight instruments were used to measure, as a function of time, the following variables: indicated airspeed, pressure altitude, normal acceleration, and pitching and rolling velocities. The head used for the measurement of airspeed was mounted on a boom extending 4.2 feet ahead of the leading edge and located 2.1 feet inboard of the left wing tip. The installation was calibrated for position error. Indicated airspeed as used in this report is defined by the following formula by which standard airspeed meters are calibrated:

$$V_i = 1703\left[\left(\frac{H-p}{p_o}+1\right)^{0.286} - 1\right]^{\frac{1}{2}}$$

where

V_i correct indicated airspeed, miles per hour

H free-stream total pressure

p free-stream static pressure

p_o standard atmospheric pressure at sea level

The free-air temperatures were obtained from radiosonde observations taken during the mornings and evenings of the dates of the test flights and were checked in flight by reading a free-air-temperature indicator connected to a temperature bulb mounted under the wing of the airplane.

TEST PROCEDURE

Tests were made with the flaps and gear up, oil- and coolant-duct shutters closed, power off, and with the propeller in the constant-speed high-pitch position.

Gradual stalls were made in turns at speeds throughout the available range at 5,100, 8,800, 13,500, 17,500, 21,300, 24,500, 29,400, and 32,300 feet pressure altitude. Stalls of varying degrees of abruptness were made at selected speeds and altitudes throughout the available Mach and Reynolds number range. With a few exceptions, all stalls were made within 300 feet of the listed altitude and 3 miles per hour of the listed speeds.

RESULTS AND DISCUSSION

The maximum lift coefficient obtainable by an airplane in flight may be limited by uncontrolled-for motions or very severe buffeting. This point is emphasized because the maximum lift coefficient obtainable in flight is determined not only by the maximum lift coefficient as measured in a wind tunnel, but also by the controllability when near the stall. The test airplane usually pitched down and rolled somewhat at the stall. At the highest Mach numbers tested, however, the airplane rolled suddenly to the right at the stall. These control characteristics at the stall may have prevented the attainment of the ultimate maximum lift coefficient that would be measured in a wind tunnel. Buffeting severe enough to prevent the maximum lift coefficient from being attained was not encountered.

In computing lift coefficients, the lift was assumed equal to the normal force WA_Z; it was estimated that the error involved in this assumption was less than 3 percent.

$$C_{L_{max}} = \frac{WA_Z}{qS}$$

where

$C_{L_{max}}$ maximum lift coefficient

W weight of the airplane, pounds

A_Z normal acceleration factor, the ratio of the net aerodynamic force along the airplane Z-axis at the stall (positive when directed upward), to the weight of the airplane

S wing area, square feet

q dynamic pressure at the stall, pounds per square foot

Gradual Stalls

The maximum lift coefficient obtainable in gradual stalls is the quantity most similar to the steady-state maximum lift coefficient measured in wind tunnels. As such, it represents the maximum useful lift coefficient and is of importance in estimating the landing speed and the turning ability of an airplane.

Curves of the maximum normal acceleration factor obtained in gradual stalls, corrected to an airplane weight of 8000 pounds, are plotted as a function of indicated airspeed at each test altitude in figure 4 and are cross-plotted as a function of altitude for constant indicated airspeed in figure 5. These curves show that, at low speeds, the maximum normal acceleration factor decreases with an increase in altitude. At high speeds and altitudes, however, a critical point is reached beyond which the maximum normal acceleration factor increases with an increase in altitude. The reasons for these variations will be shown in the subsequent sections of the report.

Effect of Mach number on the maximum lift coefficient.- The variation with Mach number of the maximum lift coefficient obtainable in gradual stalls at each test altitude is shown in figure 6. At each altitude the maximum lift coefficient decreases to 0.90 as the Mach number increases to 0.49 and then increases with further increases of Mach number reaching a value of 1.09 at a Mach number of approximately 0.66. There is an indication of a tendency to peak at this value, which is similar to the tendency previously observed in wind-tunnel tests. (See reference 2.) In the low Mach number region, the maximum lift coefficient decreases markedly with increases of altitude; whereas at higher Mach numbers the effects of altitude become very small.

The data shown in figure 6 are replotted in figure 7 to show the variation of the maximum lift coefficient with Mach number for constant Reynolds numbers. For all except the 6,500,000 and 8,000,000 Reynolds number lines the maximum lift coefficient decreases almost linearly with increases of Mach number M at a rate $\frac{\delta C_{L_{max}}}{\delta M}$ of approximately -1.75. This variation may be caused,

as described in reference 2, by the thickening of the boundary layer which reduces the circulation or by the earlier separation resulting from the steeper adverse pressure gradients associated with larger Mach numbers.

Theoretical computations made in reference 3 indicated that the wing tip and root airfoil sections have critical Mach numbers of 0.44 and 0.40, respectively, at a lift coefficient of 0.90. It appears, therefore, that the increase in the maximum lift coefficient is caused by an increase in the chordwise extent of the low-pressure region on the upper surface following the formation of a supersonic velocity region. This phenomenon has previously been noted in reference 2. In order to illustrate the point, pressure distributions from reference 2 for an NACA 16-515 airfoil at an angle of attack of 11° at Mach numbers of 0.40, 0.55, and 0.60 are presented in figure 8. These pressure distributions show that, as the speed is increased from subcritical speeds to moderately supercritical speeds, the upper-surface pressure-distribution changes from one with a sharp negative pressure peak followed immediately by a steep adverse pressure gradient to one with a lower, more rounded, negative pressure peak with the steep adverse-pressure-gradient region of the shock wave moving rearward with increases of Mach number. While this phenomenon is occurring on the upper surface, the lower-surface pressure distribution is remaining essentially unchanged, thereby accounting for the gain in lift. Unpublished data on file at Ames Aeronautical Laboratory show that similar changes in pressure distribution occur on an NACA 66,2-215 airfoil which is very similar to that used on the test airplane. As the critical Mach number is first exceeded, the rounded pressure peak produces a higher maximum lift coefficient, but as the Mach number is increased further, the loss in lift due to the decreasing values of the limit negative pressure coefficients (a concept presented in reference 4) finally overcomes the increase due to the rearward movement of the shock wave.

Figure 9 presents data obtained from reference 1 showing the variation of the maximum lift coefficient with Mach number for constant Reynolds numbers for power-off gradual stalls of the airplane equipped with the conventional wing. This airplane is similar to the test airplane except for the wing sections. The wing of the airplane described in reference 1 consists of NACA conventional sections tapering from an NACA 0015 at the wing root to an NACA 23009 at the tip. It should be noted that the curves for the airplane with the conventional wing are slightly different from those originally presented in reference 1, due to the correction of some small errors. A comparison of the curves of figure 9 with the corresponding data for the test airplane reveals that the character of the variation of the maximum lift coefficient with Mach number differs greatly for the two airplanes. In contrast to the variations previously noted for the test airplane, the maximum lift coefficient of the airplane of reference 1 decreased steadily with increasing of Mach number throughout the entire Mach number range tested. Unpublished airfoil

data on file at Ames Aeronautical Laboratory indicate that this characteristic is typical of conventional airfoils and that the large chordwise extension of the low-pressure region at moderately supercritical Mach numbers illustrated in figure 8 does not occur on conventional airfoils.

A comparison of the data obtained at low Mach numbers reveals that the maximum lift coefficient of the airplane with the conventional wing is greater than that of the test airplane. At supercritical Mach numbers, however, the maximum lift of the conventional wing was much less than that of the low-drag wing.

Effect of Reynolds number on the maximum lift coefficient.- Figure 10, a cross plot of figure 7, shows that as the Reynolds number is increased at constant Mach numbers the maximum lift coefficient at first remains nearly constant; but when a critical Reynolds number is reached, the maximum lift coefficient increases rapidly to a higher value and then remains nearly constant again with further increases of Reynolds number. The rapid increase of the maximum lift coefficient has been shown in reference 5 to be associated with the change from laminar separation to turbulent separation. Figure 11 shows that the critical Reynolds number (Reynolds number at which the maximum lift coefficient starts its rapid increase) increases nearly linearly with Mach number. There are two effects which could cause such a variation of critical Reynolds number with Mach number. One is the increase of kinematic viscosity in the boundary layer of a compressible fluid due to aerodynamic heat which causes the ratio of the local Reynolds number (based on boundary-layer conditions) to the free-stream Reynolds number to diminish as shown in reference 6. Accordingly, as shown by references 7 and 8, a larger free-stream Reynolds number would be necessary to reach the local critical Reynolds numbers required for transition from a laminar to a turbulent boundary layer. A second possibility is due to the fact that increasing the Mach number in the subcritical range has effects on the upper-surface pressure distribution similar to that of decreasing the airfoil thickness. The pressure peaks become sharper and the adverse pressure gradients become steeper. Decreasing the airfoil thickness is shown in reference 5 to increase the critical Reynolds number, hence similar effects due to increasing Mach number would be anticipated.

As the Mach number increases, the effects of Reynolds number on the maximum lift coefficient decrease until, at supercritical Mach numbers, no effects are apparent. The Reynolds number effects are probably suppressed in the supercritical Mach number region because the separation point may be determined entirely by the position of the shock wave rather than by the gradual growth of the boundary layer that occurs in subcritical flow.

Figure 12 for the airplane of reference 1 indicates that the Reynolds number effects on the maximum lift coefficient are of a

similar nature to those determined for the test airplane. As the Mach number is increased, the effects of Reynolds number decrease until at a Mach number of 0.60, no effects are discernible. Although not included in the test range, it appears from the shape of the curves that the critical Reynolds number increases with increases of Mach number in a manner similar to that observed for the test airplane.

The results of these tests indicate the necessity for considering both the individual and interrelated effects of Mach and Reynolds numbers in the prediction from wind-tunnel-model data of the maximum lift coefficient of an airplane flying at subcritical Mach numbers. This becomes of special importance in the prediction of landing speed. At speeds greater than the critical Mach number, however, the maximum lift coefficient is shown to be independent of Reynolds number over the range of the tests (9,000,000 to 16,500,000).

<u>Calculated minimum radius of turn.</u>- The effects of the variation of the maximum lift coefficient with Mach and Reynolds numbers on the maneuvering characteristics of the test airplane are illustrated by the curves of figure 13 showing the calculated minimum radius of a properly banked, power-off, horizontal turn at various altitudes plotted as a function of indicated airspeed and that computed by using a low Mach number value of the maximum lift coefficient (1.30).

A comparison of the two curves for each altitude shows the large detrimental effects on the turning ability of the test airplane that result from the decrease of the maximum lift coefficient with increases in Mach number. This comparison indicates the necessity for considering Mach number effects in estimating the turning performance of airplanes, especially at high altitudes.

Abrupt Stalls

Although the higher lift obtained in abrupt stalls is not a useful quantity in increasing the maneuverability or decreasing the landing speed of an airplane, it is considered, at present, in designing airplanes to withstand loads imposed by vertical gusts and by abrupt maneuvers. Although the tests were made by stalling the airplane in very abrupt pull-ups, similar maximum lift characteristics would be anticipated in stalls produced by flying into a vertical gust because both phenomena are caused essentially by the lag of the flow separation following a sudden increase in the angle of attack. The effects of pitching velocity and Mach and Reynolds numbers on the maximum lift coefficient will be discussed in the following sections.

<u>Effects of pitching velocity on the maximum lift coefficient.</u>- The effects of pitching velocity on the maximum lift coefficient are

shown in figure 14 where the maximum lift coefficient is plotted for various speeds and altitudes as a function of the excess pitching angle per chord length of travel. This parameter is

$$\frac{\Delta q \bar{c}}{V_T}$$

where

Δq excess pitching velocity (the difference between the actual pitching velocity and that required to maintain the airplane in a steady turn or pull-up at a constant angle of attack corresponding to the speed, altitude, normal acceleration, and attitude of the airplane at the time of the stall), degrees per second

$\bar{c}$ mean aerodynamic chord, feet

V_T true airspeed at the time of the stall, feet per second

Use of this parameter to aid in the general application of these data is justified in reference 9 where it is pointed out that it represents the condition for dynamic similitude of unsteady acceleration forces.

Figure 14 shows that the maximum lift coefficient increases nearly linearly with rate of pitching until a limiting value of the maximum lift coefficient is reached which is unaffected by further increases in pitching velocity.

Effect of Mach and Reynolds numbers on the maximum lift coefficient.- The limiting value of the maximum lift coefficient obtainable in abrupt stalls is plotted as a function of Mach number and altitude in figure 6 and is shown to decrease rapidly with increases of Mach number but to be independent of the altitude and, consequently, the Reynolds number. As a result of theoretical calculations indicating that the critical Mach number of the wing is exceeded at lift coefficients below the limiting value obtained in the abrupt stalls, it appears that the maximum lift coefficient is limited by the limit pressure coefficients or by boundary-layer separation induced by the compression shock wave rather than by the normal growth and separation of the boundary layer. The maximum lift coefficient, therefore, becomes relatively independent of the lag of the flow separation and remains nearly constant with further increases of pitching velocity.

The trends of the curves shown in figure 6 suggest that the maximum lift coefficient obtainable in abrupt stalls may approach that obtainable in gradual stalls at a Mach number of approximately 0.66. Such an extrapolation appears plausible because, in both gradual and abrupt stalls at these Mach numbers, the maximum lift coefficient is becoming limited by the reduction in limit pressure coefficient.

Comparison With Calculated Limit Lift Coefficient

The limit lift coefficient, a concept introduced in reference 4, is defined as the lift coefficient at which potential flow ceases to exist (viz, the lift coefficient at the so-called compressibility burble) and is shown to be in agreement with the maximum lift coefficients measured in abrupt stalls of a P-47C-1 airplane.

The limit lift coefficient was computed for the test airplane by the method outlined in reference 4. Figure 15 shows a comparison of this calculated limit lift coefficient with the maximum lift coefficient obtainable in the gradual and abrupt stalls. At Mach numbers less than 0.61, the curve shows that the calculated limit lift coefficient is less than the maximum lift coefficient measured in abrupt stalls, and that this limit lift coefficient decreases with increases in Mach number. At a Mach number of 0.61, however, the calculated limit lift coefficient increases suddenly, reaching, at a Mach number of 0.64, a value slightly greater than the maximum lift coefficient measured in gradual stalls at that Mach number. With further increases in Mach number, the limit lift coefficient decreases abruptly reaching a value of zero at a Mach number of approximately 0.69.

Although the value of the limit lift coefficient is in fair agreement with the maximum lift coefficient at Mach numbers less than 0.65, the temporary rise in the limit lift coefficient at a Mach number of 0.61 is merely a coincidence; and the abrupt drop at a Mach number of 0.64 is not indicated by the trend of the flight data, nor is it shown by unpublished wind-tunnel data (on file at Ames Aeronautical Laboratory) on similar sections tested to much higher Mach numbers. The discrepancy is traceable, in this instance, to rather large differences between experimental high Mach number pressure distribution and those predicted by the method of reference 4. The sudden increase of the calculated limit lift coefficient at a Mach number of 0.61 occurs when the angle of attack decreases to the value at which the minimum-pressure-peak position moves rapidly from near the leading edge to the 60-percent-chord station. With further increases of Mach number, the negative lift on the lower surface increases rapidly until, at a Mach number of 0.685, the calculated limit lift coefficient approaches zero or an indeterminate value.

CONCLUSIONS

From tests of the maximum lift obtainable in gradual and abrupt stalls of the test airplane, the following conclusions have been made:

1. A limiting value of the maximum lift coefficient obtainable in very abrupt stalls was found which decreased with increases in Mach number and was independent of Reynolds number.

2. Reynolds number had less effect on the maximum lift coefficient obtainable in gradual stalls at high Mach numbers than at low Mach numbers, no effects being apparent at Mach numbers greater than 0.50.

3. The increase in the maximum lift coefficient due to Reynolds number occurred at higher values of Reynolds number at high Mach numbers than at low Mach numbers.

4. The maximum lift coefficient was affected by compressibility at Mach numbers as low as 0.15.

5. The maximum lift coefficient obtainable in gradual stalls decreased nearly linearly with increases in Mach number until a minimum value of 0.90 was reached at a Mach number of 0.49, and then increased with further increases in Mach number until a value of 1.09 was reached at a Mach number of 0.66.

6. At low Mach numbers the maximum lift of the airplane with the conventional wing was greater than that of the test airplane equipped with the low-drag wing. At moderately supercritical Mach numbers, however, the maximum lift of the low-drag wing was much greater than that of the conventional wing.

Ames Aeronautical Laboratory,
National Advisory Committee for Aeronautics,
Moffett Field, Calif., July 6, 1945.

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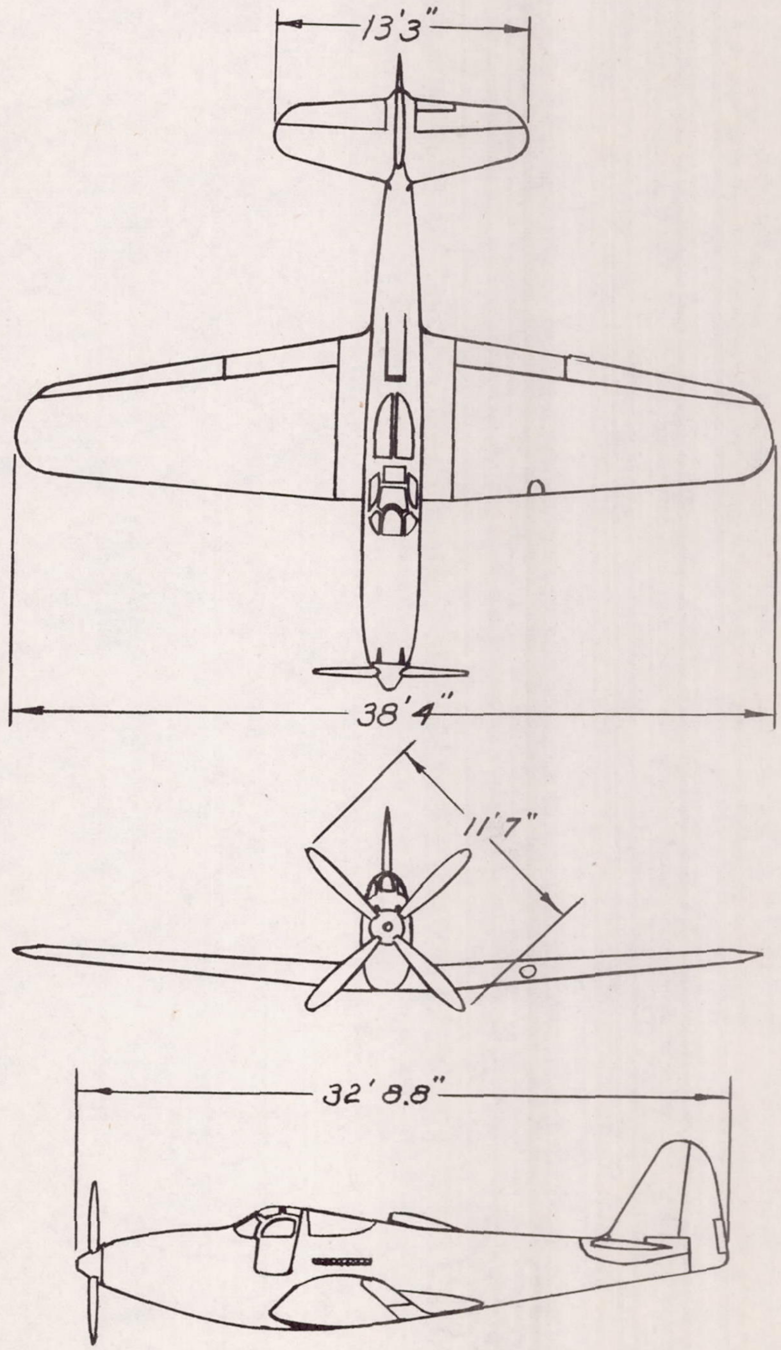

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FIGURE 1.- THREE VIEW DRAWING OF THE TEST AIRPLANE.

Figure 2.— The test airplane as instrumented for test flights.

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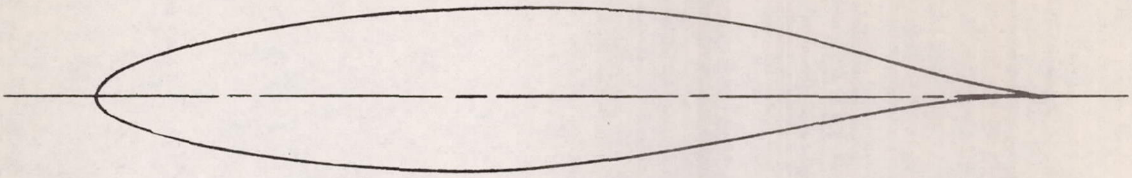
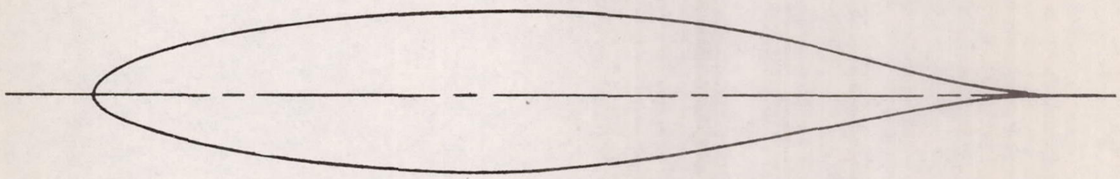


NACA 66,2X-216
TIP AIRFOIL



NACA 66,2X-116
ROOT AIRFOIL

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FIGURE 3.—PROFILES OF TIP AND ROOT AIRFOIL SECTIONS OF THE TEST AIRPLANE

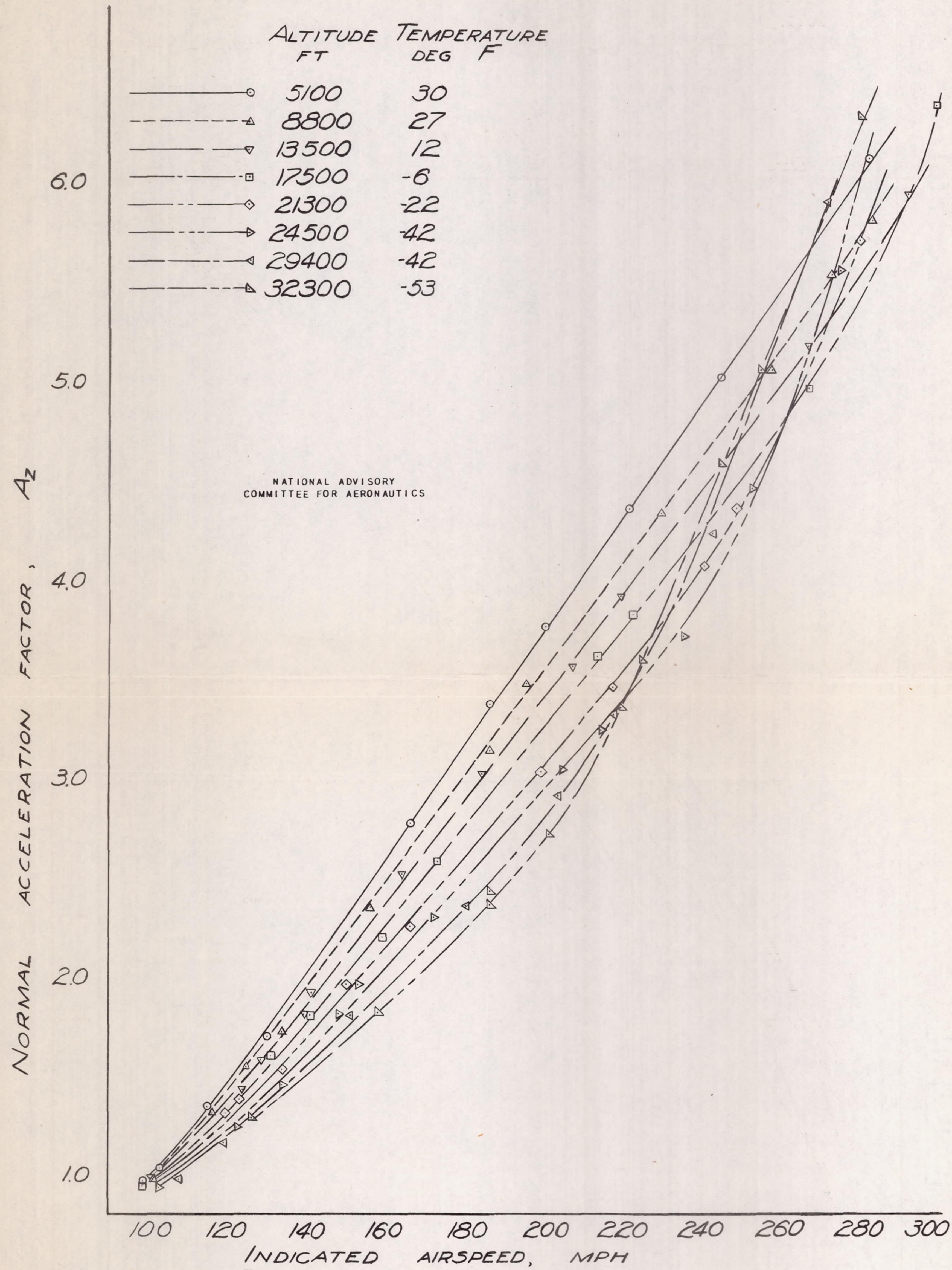

FIGURE 4.— MAXIMUM NORMAL ACCELERATION FACTOR OBTAINABLE IN GRADUAL STALLS PLOTTED AS A FUNCTION OF INDICATED AIRSPEED FOR VARIOUS ALTITUDES. TEST AIRPLANE.

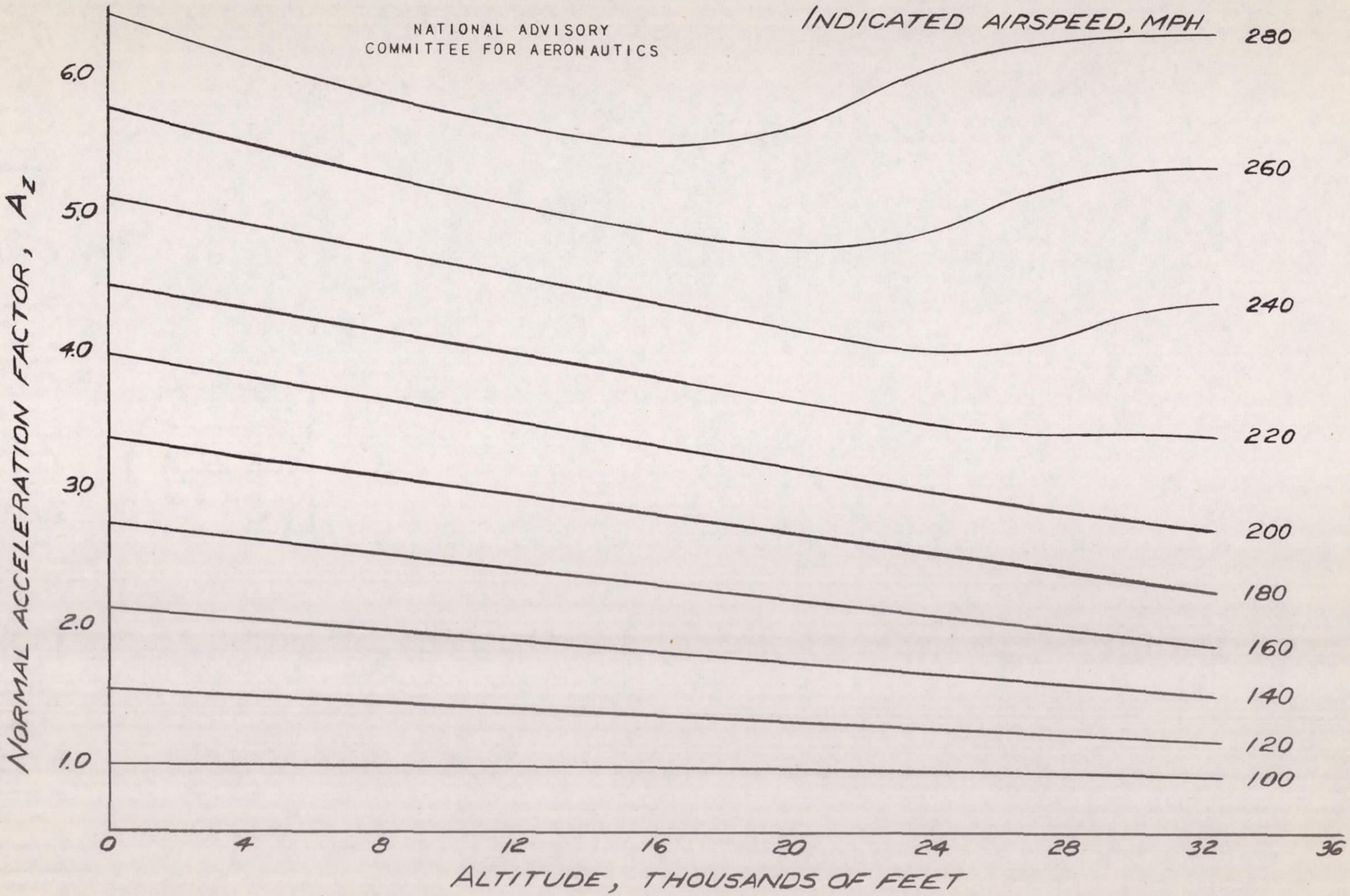

FIGURE 5.— MAXIMUM NORMAL ACCELERATION FACTOR OBTAINABLE IN GRADUAL STALLS PLOTTED AS A FUNCTION OF ALTITUDE FOR VARIOUS AIRSPEEDS. TEST AIRPLANE.

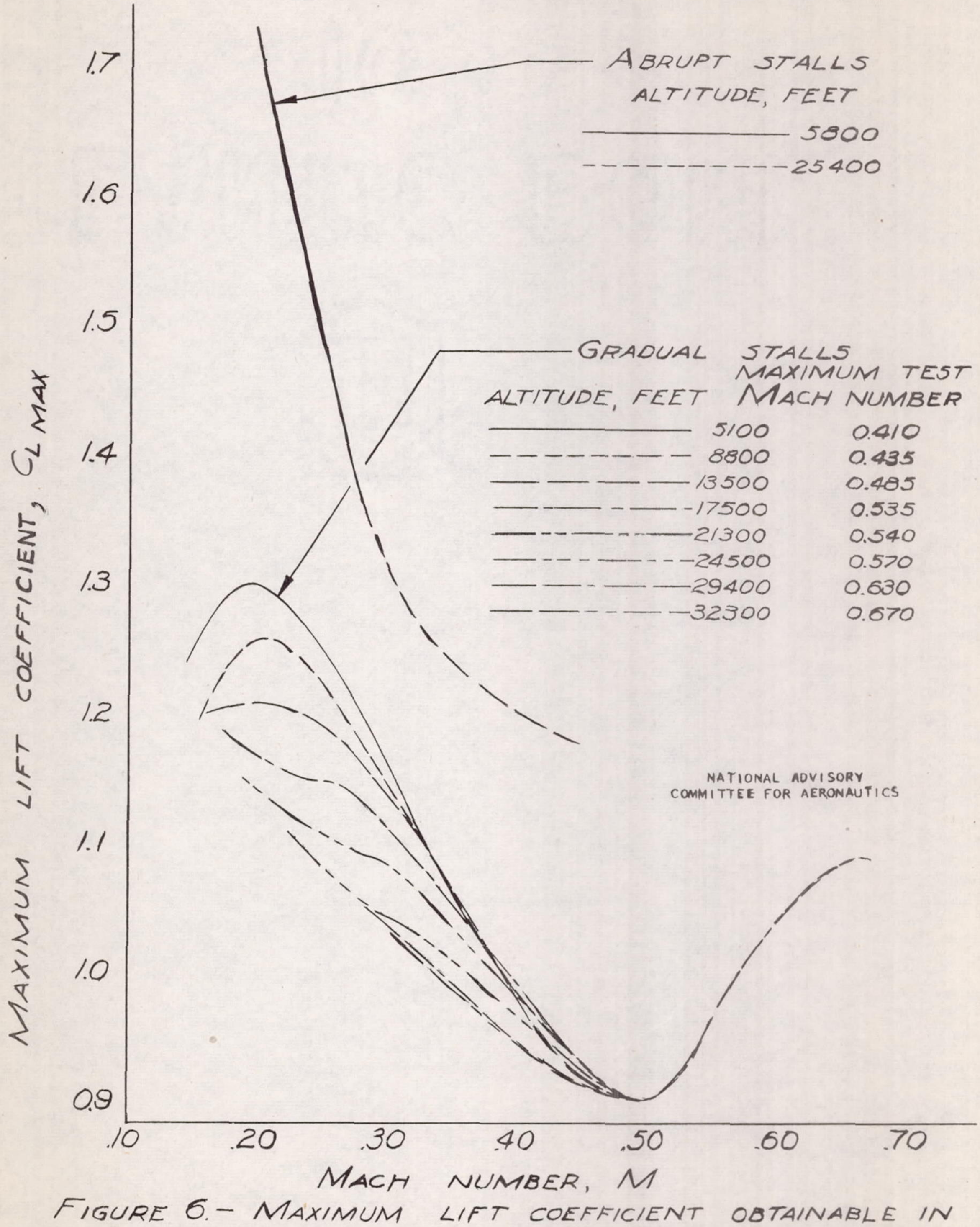

FIGURE 6.– MAXIMUM LIFT COEFFICIENT OBTAINABLE IN GRADUAL AND ABRUPT STALLS PLOTTED AS A FUNCTION OF MACH NUMBER FOR VARIOUS ALTITUDES. TEST AIRPLANE.

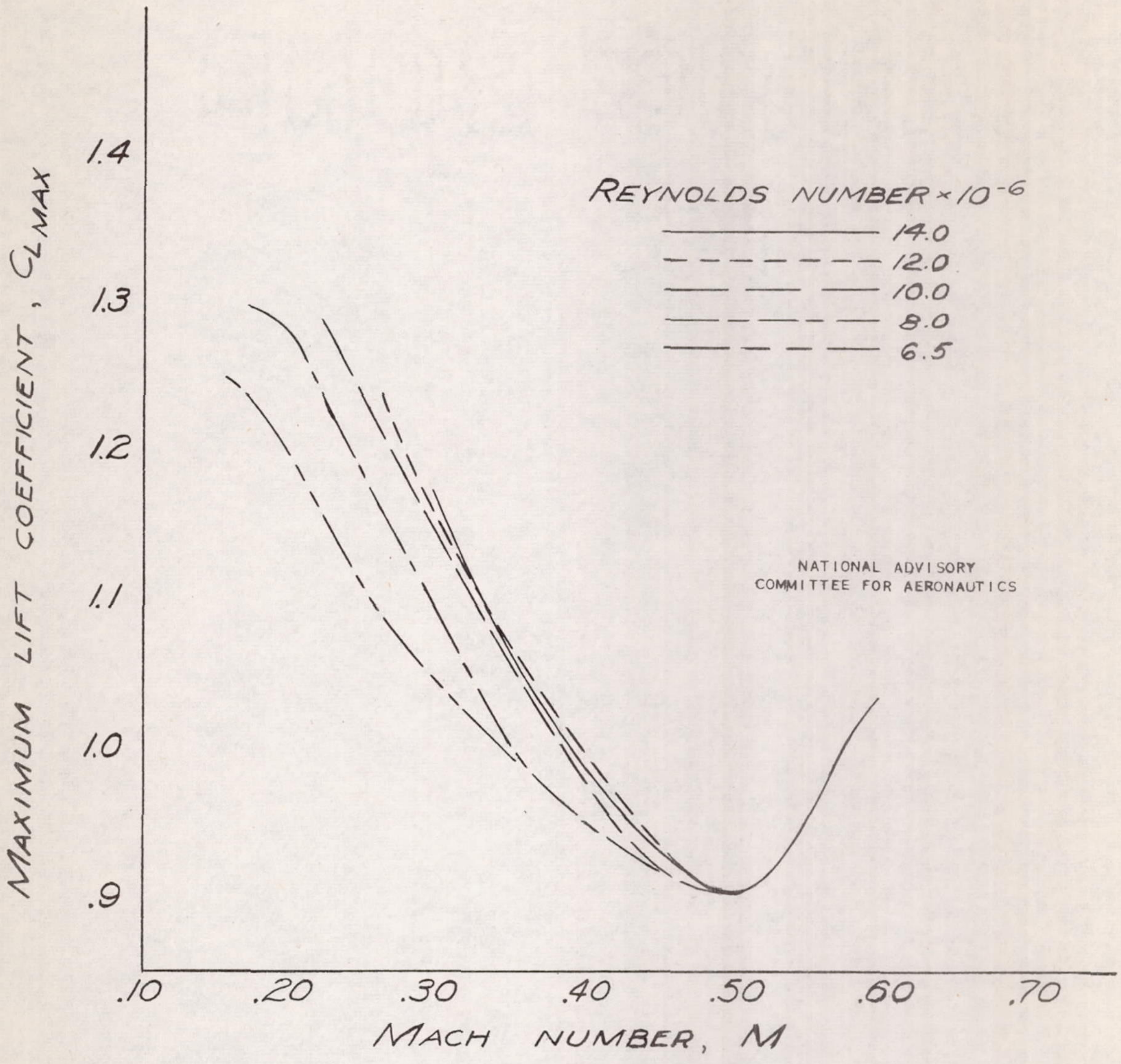

FIGURE 7.— VARIATION OF MAXIMUM LIFT COEFFICIENT OBTAINABLE IN GRADUAL STALLS WITH MACH NUMBER FOR VARIOUS REYNOLDS NUMBERS. TEST AIRPLANE.

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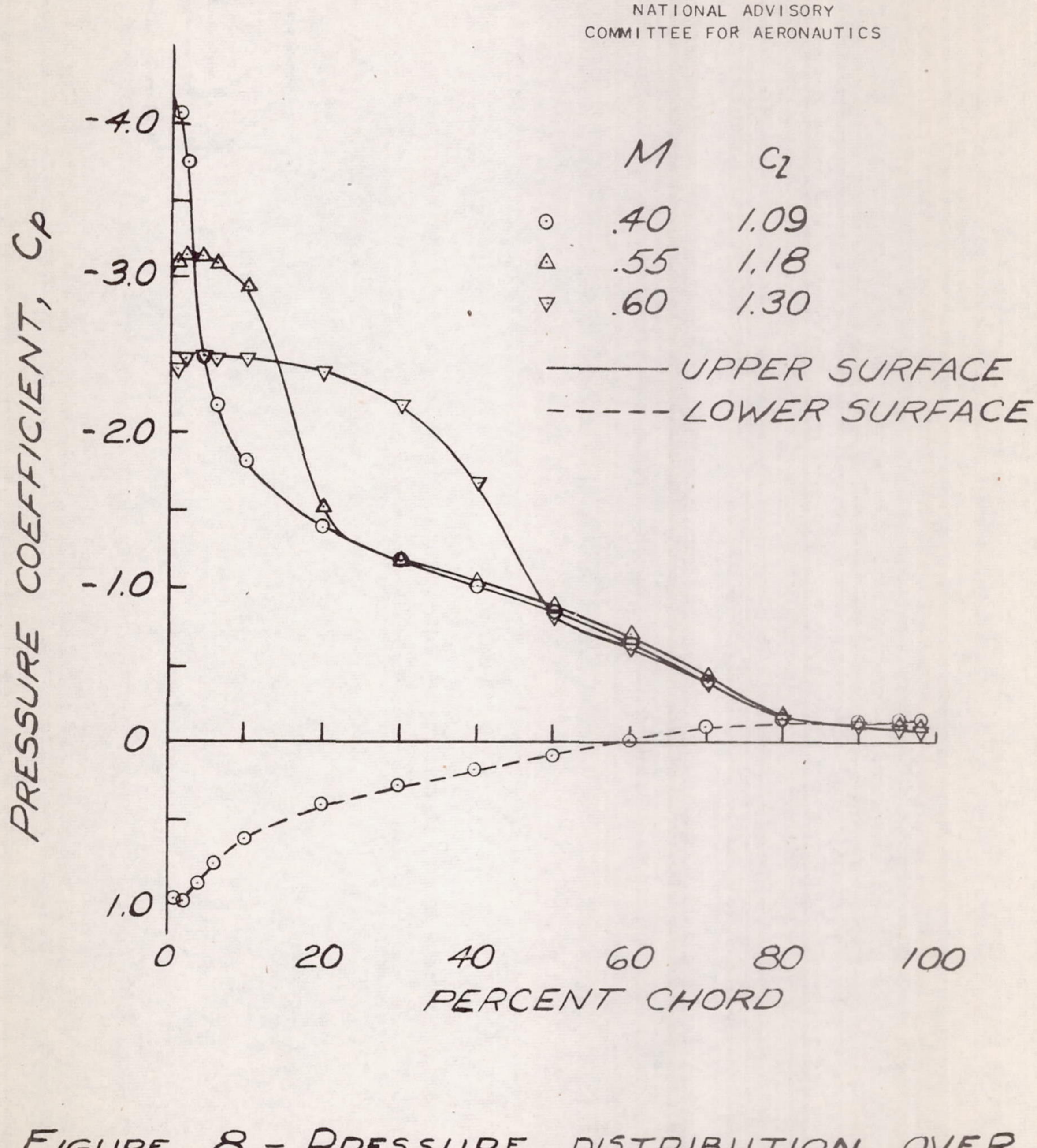

FIGURE 8.- PRESSURE DISTRIBUTION OVER THE NACA 16-515 AIRFOIL AT AN ANGLE OF ATTACK OF 11° AT MACH NUMBERS OF 0.40, 0.55 AND 0.60. (REFERENCE 3.)

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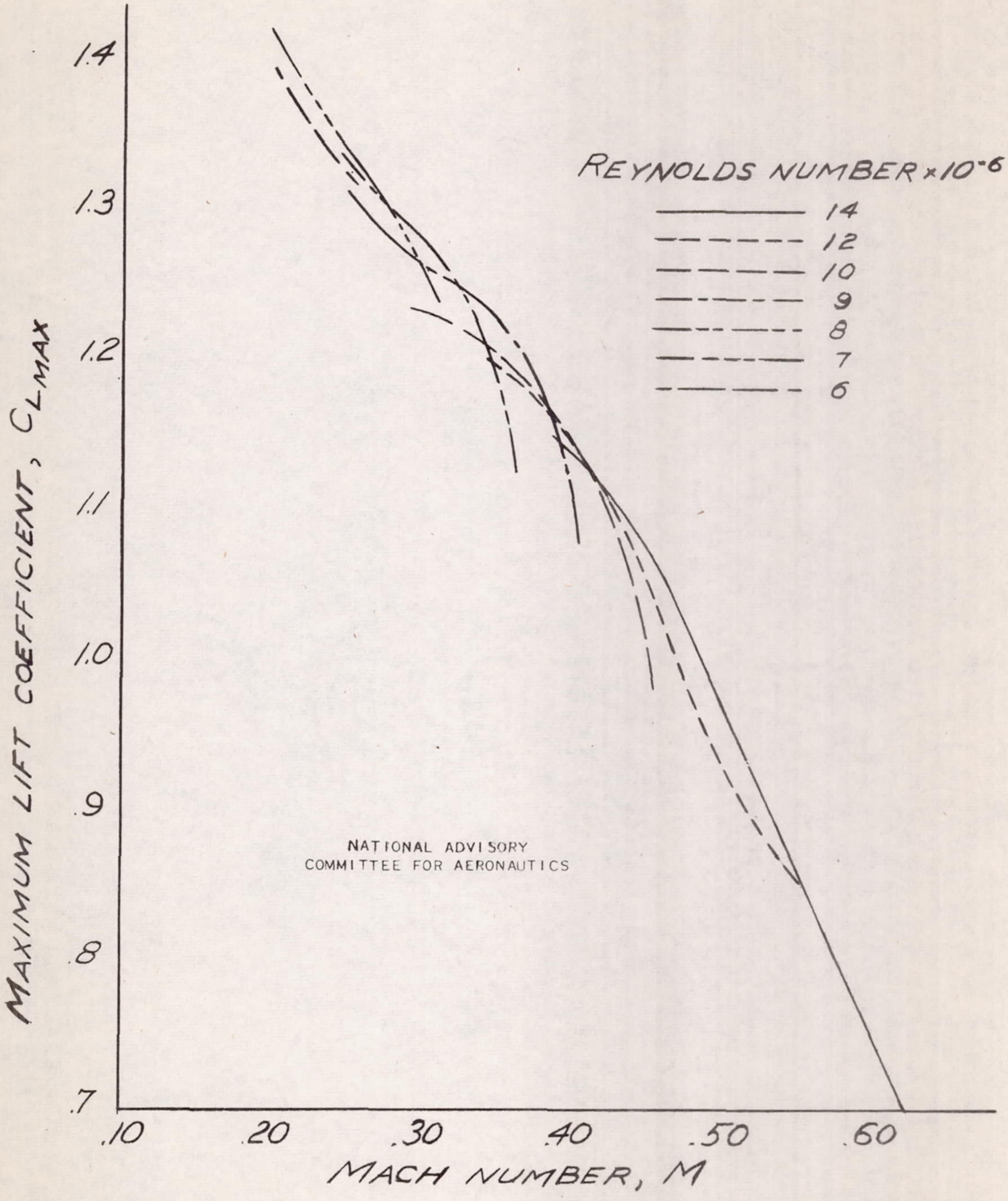

FIGURE 9. — VARIATION OF MAXIMUM LIFT COEFFICIENT OBTAINABLE IN GRADUAL STALLS WITH MACH NUMBER FOR VARIOUS REYNOLDS NUMBERS. AIRPLANE WITH CONVENTIONAL WING. (REFERENCE 1).

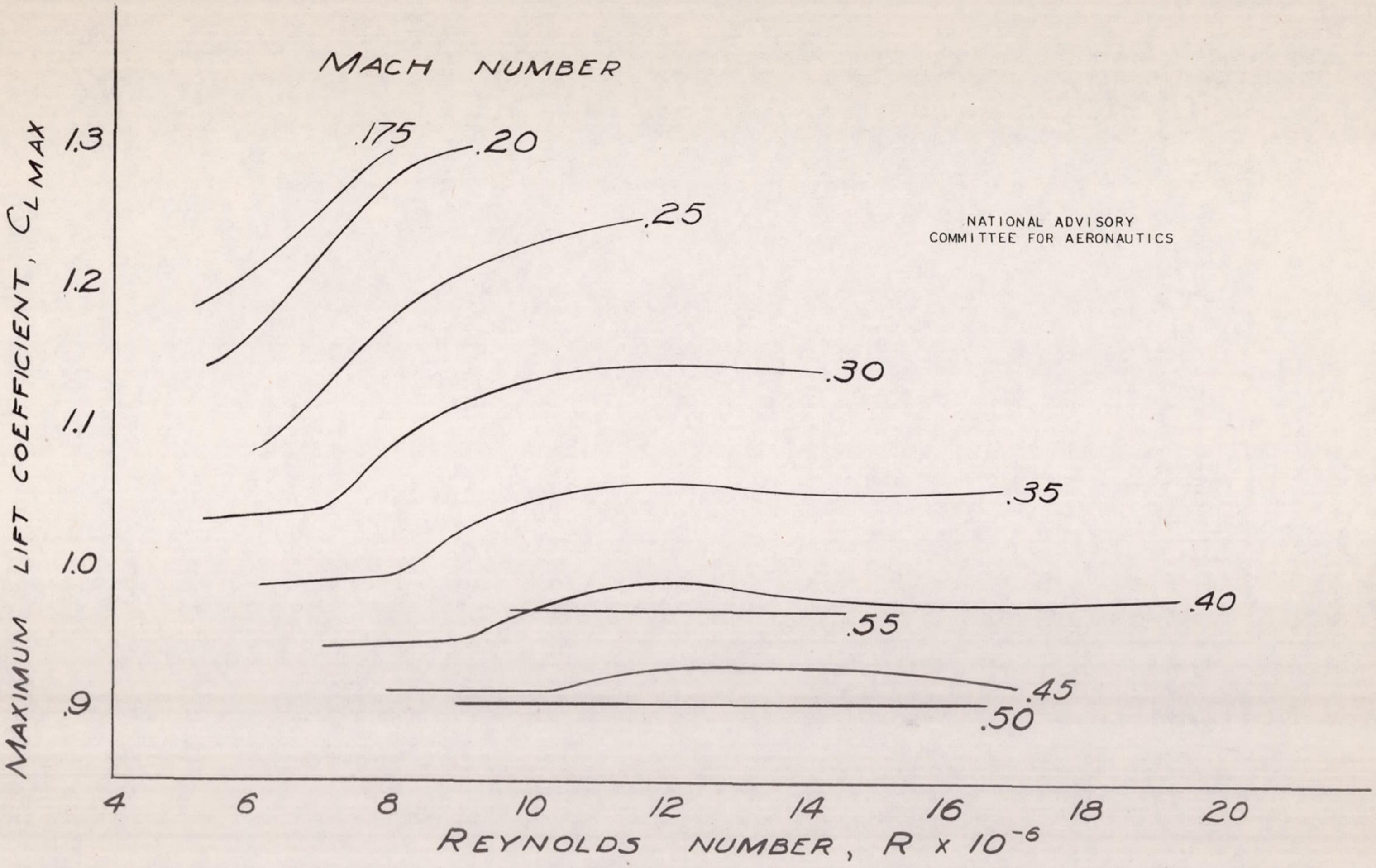

FIGURE 10. — VARIATION OF MAXIMUM LIFT COEFFICIENT OBTAINABLE IN GRADUAL STALLS WITH REYNOLDS NUMBER FOR VARIOUS MACH NUMBERS. TEST AIRPLANE.

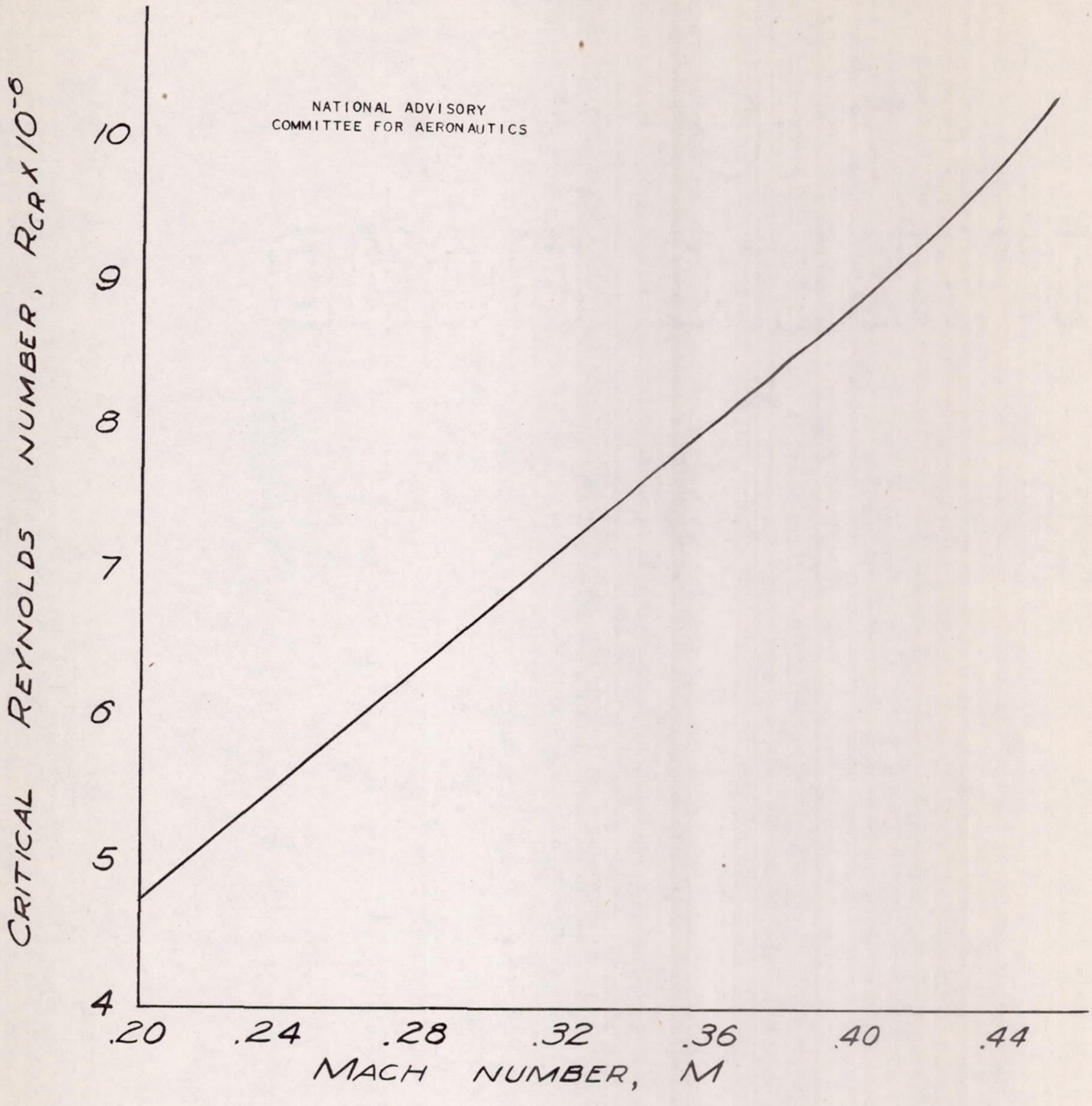

FIGURE 11.— VARIATION OF CRITICAL REYNOLDS NUMBER WITH MACH NUMBER. TEST AIRPLANE.

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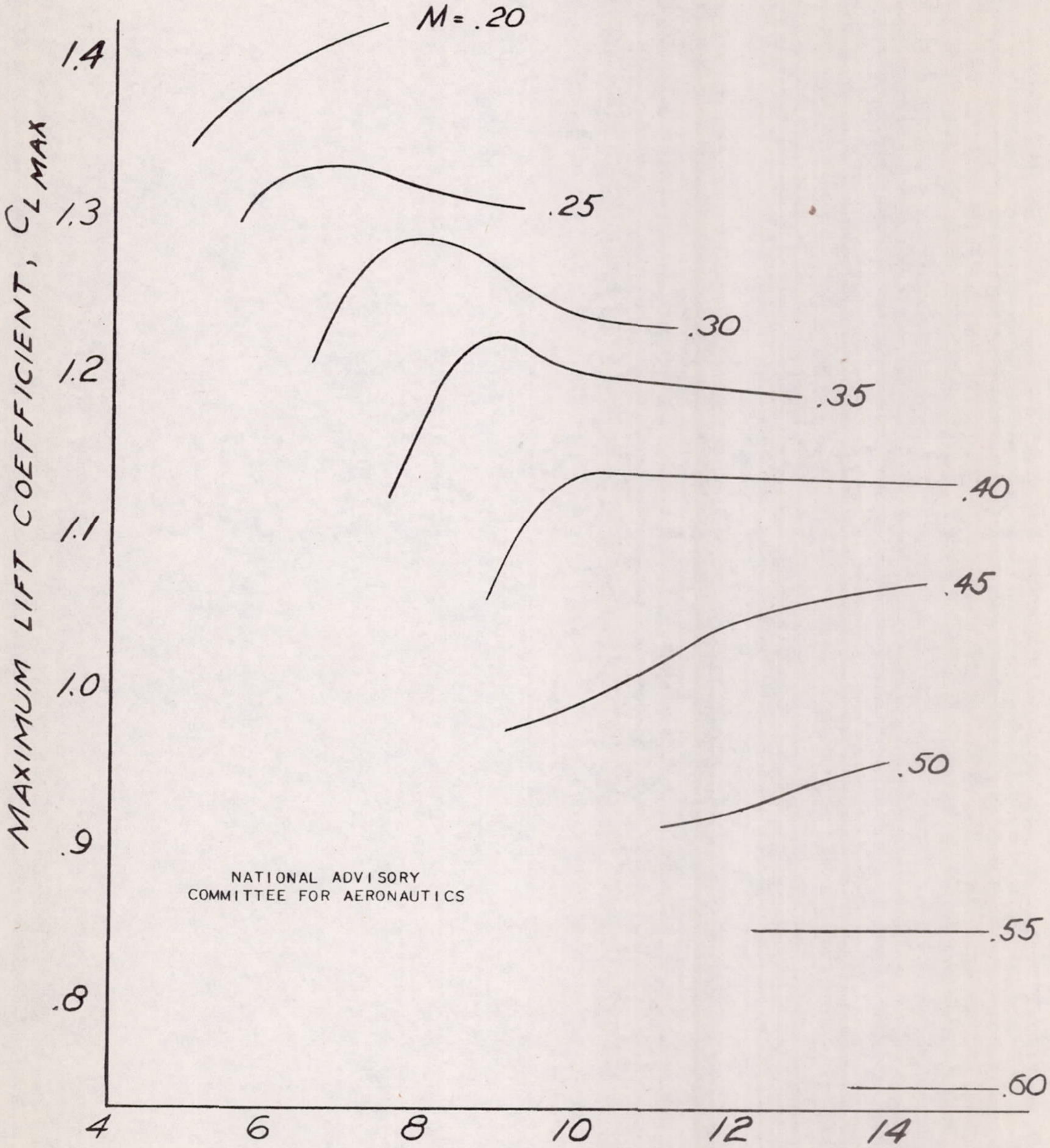

FIGURE 12 - VARIATION OF MAXIMUM LIFT COEFFICIENT OBTAINABLE IN GRADUAL STALLS WITH REYNOLDS NUMBER FOR VARIOUS MACH NUMBERS. AIRPLANE WITH CONVENTIONAL WING. (REFERENCE 1).

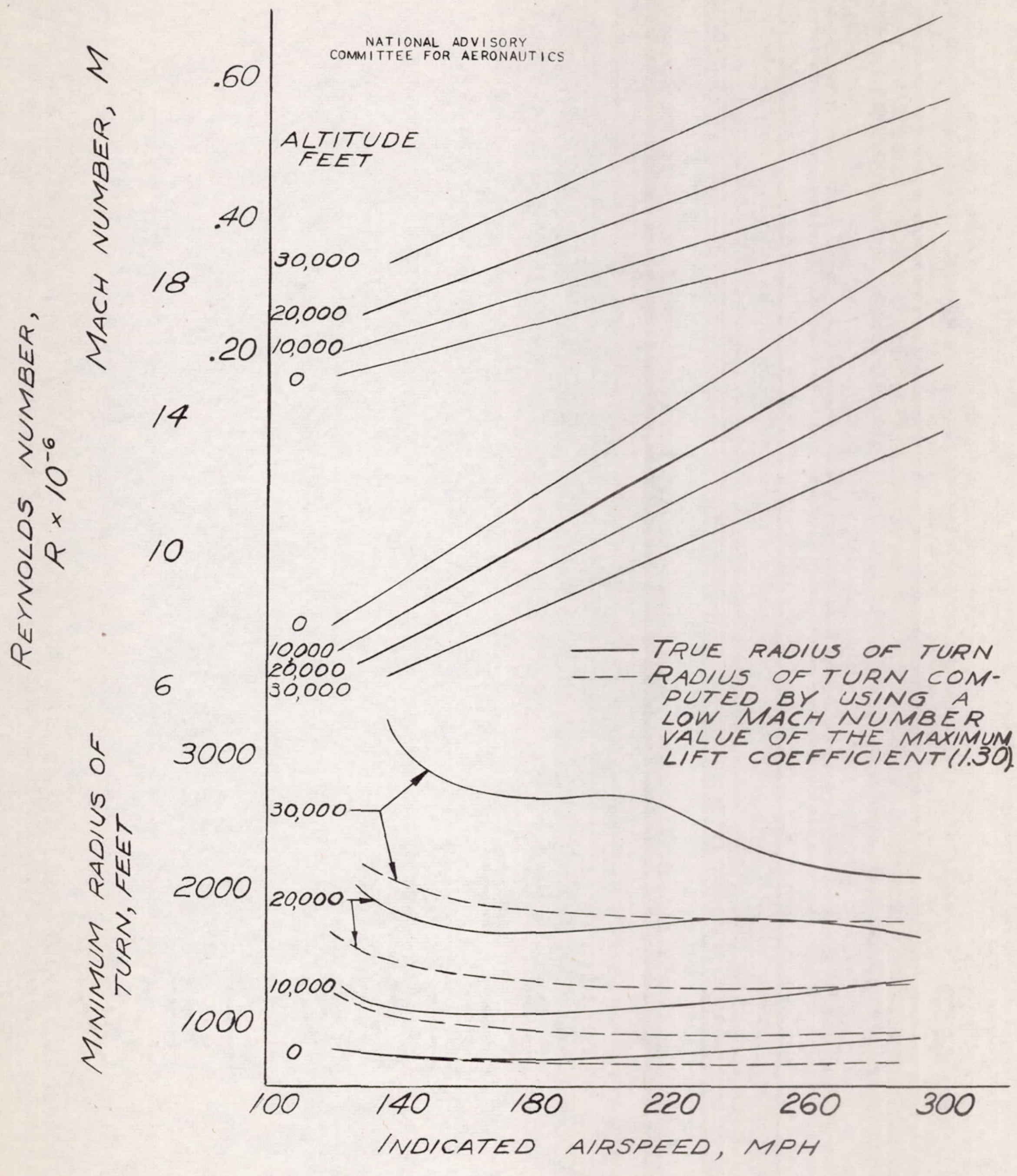

FIGURE 13.— CALCULATED MINIMUM RADIUS OF TURN PLOTTED AS A FUNCTION OF AIRSPEED FOR VARIOUS ALTITUDES. TEST AIRPLANE.

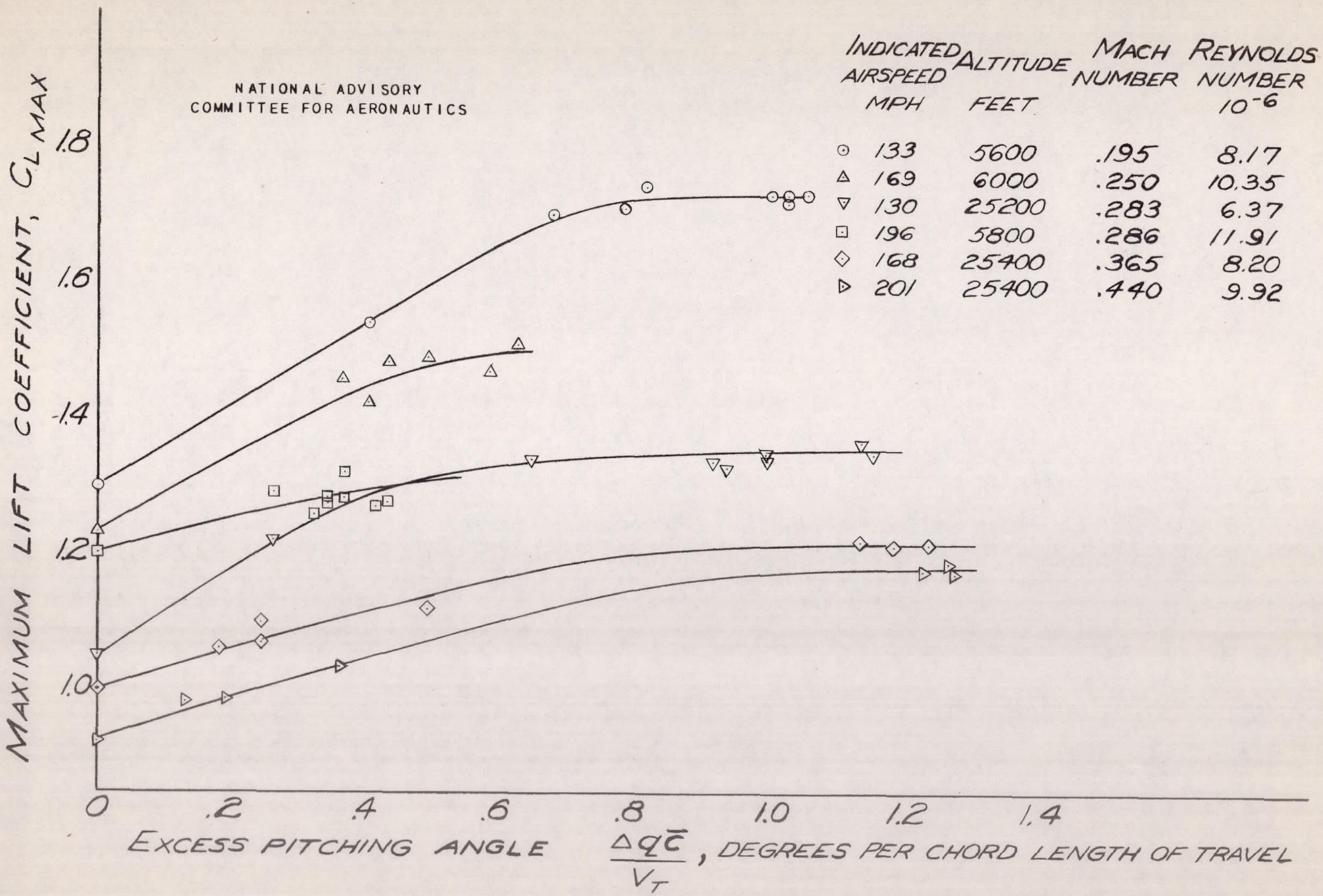

FIGURE 14.— MAXIMUM LIFT COEFFICIENT OBTAINABLE IN ABRUPT STALLS PLOTTED AS A FUNCTION OF EXCESS PITCHING ANGLE FOR VARIOUS AIRSPEEDS AND ALTITUDES. TEST AIRPLANE.

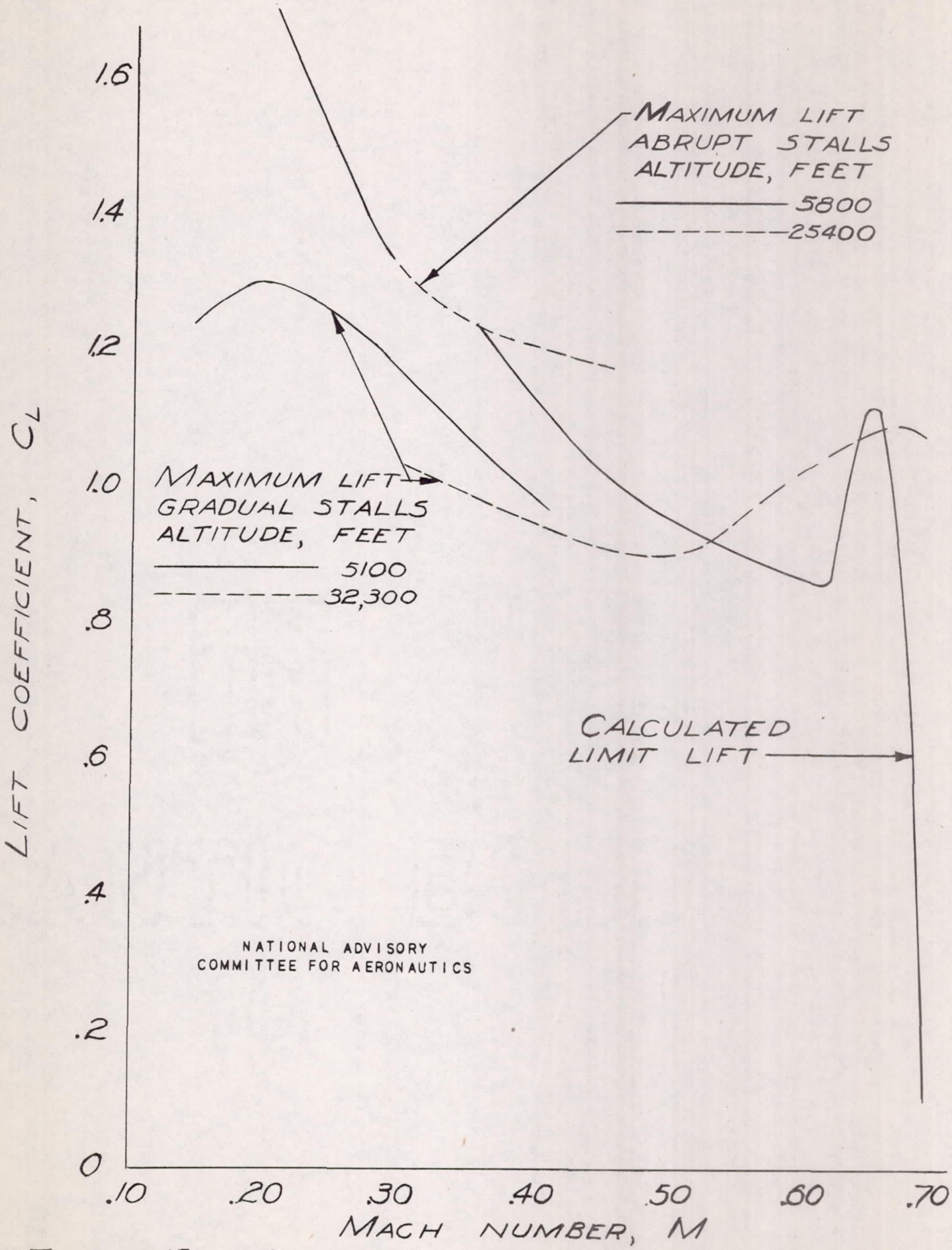

FIGURE 15 - MAXIMUM LIFT COEFFICIENT OF THE TEST AIRPLANE COMPARED WITH CALCULATED WING LIMIT LIFT COEFFICIENT.